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RESEARCH MEMORANDUM

ANALYTICAL AND EXPERIMENTAL INVESTIGATION OF INLET-ENGINE

MATCHING FOR TURBOJET-POWERED AIRCRAFT

AT MACH NUMBERS UP TO 2.0

By Carl F. Schueller and Fred T. Esenwein

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NATIONAL ADVISORY COMMITTEE
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SUMMARY

The problems associated with the design of high performance inlets suitable for a turbojet-powered aircraft operating from Mach number 0 to 2.0 are discussed herein. The results of an analysis of inlet - turbojet-engine matching for a range of Mach numbers to 2.0 are substantiated by an experimental investigation conducted in the NACA Lewis 8- by 6-foot supersonic wind tunnel at Mach numbers of 0, 0.63, and 1.5 to 2.0. The model included two ramp-type side inlets mounted symmetrically about the vertical center line of a fuselage having a modified triangular cross section. Scaled internal ducts extending to the face of the engine compressor and ram-type boundary-layer-removal scoops were included in the one-quarter-scale model. The research was conducted at Reynolds numbers from approximately 19×10^6 for a Mach number of 0.63 to 29×10^6 for supersonic Mach numbers based on the length of fuselage ahead of the inlet.

Results of the analysis indicate that the use of fixed-geometry-inlet designs in conjunction with a representative turbojet engine operating over a Mach number range from 0.60 to 2.0 will result in large performance penalties. Use of variable-geometry inlets, however, greatly reduces these penalties. Experimentally this was confirmed by investigating two inlets of different compression-ramp angles which simulated a variable geometry configuration. With complete removal of the boundary layer ahead of the inlets, total-pressure recoveries comparable with those attainable with well-designed nose inlets were obtained.

The use of blunt-inlet leading edges designed from subsonic considerations resulted in serious drag penalties at a Mach number of 2.0, whereas sharp-inlet leading edges for high performance at supersonic velocities produced large losses in thrust at take-off. These thrust penalties which are associated with the low-speed operation of the sharp-lip inlet designs can probably be avoided without impairing the supersonic performance of the inlet by the use of auxiliary inlets or blow-in doors.

INTRODUCTION

Supersonic nose inlets designed for operation at or near a specific free-stream Mach number have been evaluated experimentally for Mach numbers up to 2.0 by a number of investigators. Only limited research, however, has been conducted to evaluate the performance of inlets which are required to operate over the wide range of flight Mach numbers, altitudes, and engine air flows which are typical of turbojet-powered aircraft operating from take-off to supersonic speeds (reference 1).

An analytical and experimental investigation of side inlets for turbojet-powered aircraft operating at Mach numbers up to 2.0 was conducted at the NACA Lewis laboratory and the results are presented herein. For the analysis a two-dimensional single-oblique-shock-type inlet was considered. Performance characteristics of fixed-geometry inlets are indicated and a method of matching the inlet characteristics to the engine air-flow requirements is demonstrated.

For the experimental phase of the investigation two ramp-type semi-circular side inlets were investigated on a fuselage having a modified triangular cross section. Pressure recovery and drag data were obtained for 14° and 6° compression-ramp angles to simulate two positions of a practical variable-geometry inlet. The investigation was conducted at Mach numbers of 0.63, 1.5, 1.7, 1.9, and 2.0 for the cruise angle of attack of 3°. Additional data were obtained for the static take-off conditions. The Reynolds number based on the length of fuselage ahead of the inlets was approximately 29×10^6 for the supersonic Mach numbers and 19×10^6 for Mach number 0.63.

SYMBOLS

The following symbols are used in this report:

A   area

C_{D_f} model fore-drag coefficient based on maximum body cross-sectional area of 1.784 square feet

D   drag

F_n engine net thrust

h   height of boundary-layer scoop

M   Mach number

m   mass flow

m_2/m_1 mass-flow ratio, $\rho_2 V_2 A_2/\rho_1 V_1 A_p$

n engine speed, rpm

P total pressure

V velocity

W weight flow

γ ratio of specific heats

δ ratio of local total pressure to static pressure of NACA
      standard atmosphere at sea level (2116 lb/sq ft absolute)

δ_{bl} boundary-layer thickness

θ ratio of total temperature to static temperature of NACA
      standard atmosphere at sea level (519° R)

ρ density

Subscripts:

a   additive

c   compressor-inlet station

d   design

f   model forebody

i   inlet

p   projected area of inlet defined by leading edges of cowl and
      compression ramp

0   free stream

1   canopy station ahead of inlet

2   diffuser-discharge station

INLET-ENGINE MATCHING

The air-flow requirements of a turbojet engine can be generalized,
if Reynolds number effects are neglected, by a single curve when the
corrected air flow is plotted against the corrected engine speed. Typi-
cal generalized air-flow requirements for three current engine designs

are shown in figure 1(a). For operation at constant engine speed over the range of flight conditions, the abscissa becomes $1/\sqrt{\theta}$ and is therefore a function of only altitude and free-stream Mach number. At altitudes of 35,000 feet and above, for which the air temperature is constant, θ will be dependent on the flight Mach number alone.

The generalized engine air-flow characteristics presented in figure 1(a) can be expressed in terms of the compressor-inlet conditions as

$$\left(\frac{W\sqrt{\theta}}{\delta}\right)_c = \frac{85.3 \, M_c}{\left(1 + \frac{\gamma-1}{2}M_c^2\right)^3} A_c \tag{1}$$

and for a fixed compressor-inlet area A_c, a required schedule of compressor-inlet Mach numbers can be determined for each engine. The variation of compressor-inlet Mach number with flight Mach number is presented in figure 1(b) for engine B, which will be considered in this investigation. The compressor-inlet Mach number is different from the diffuser-discharge Mach number in this example because of the presence of the engine-accessory housing. Therefore, in order to facilitate analysis of the inlet performance, values of the diffuser-discharge Mach numbers corresponding to the compressor-inlet Mach numbers were calculated by assuming isentropic flow between these stations for the geometrical area ratio A_c/A_2 of 0.71. The required diffuser-discharge Mach numbers shown in figure 2(a) indicate that at constant altitude and engine speed the compressor will operate at only one diffuser-discharge Mach number for each free-stream Mach number. In contrast, an isolated inlet is capable of operating over a wide range of discharge Mach numbers. The inlet-engine matching problem therefore is associated with the design of inlets having high performance characteristics at the diffuser-discharge Mach number required by the engine operating conditions.

In order to obtain some idea of the necessary inlet requirements for this particular engine, the engine corrected air flow has been expressed in terms of the free-stream conditions as

$$\left(\frac{W\sqrt{\theta}}{\delta}\right)_c = \frac{A_0}{P_c/P_0} \frac{85.3 \, M_0}{\left(1 + \frac{\gamma-1}{2}M_0^2\right)^3} \tag{2}$$

where P_c/P_0 represents the pressure recovery at the face of the compressor. The resulting area requirements A_0/A_c presented in figure 2(b) for an altitude of 35,000 feet and higher indicate that a considerable variation in stream-tube area is required for operation over the Mach number range from 0 to 2.0 for the estimated schedule of inlet pressure recoveries shown by the dashed line.

Supersonic external compression inlets can be designed such that the entering stream tube is equal to the projected frontal area of the inlet by maintaining the oblique and normal shocks at the lip of the inlet. For the condition of zero spillage and an attainable schedule of pressure recovery (as shown by the dashed line in fig. 2(b)), a projected inlet-area variation of approximately 17 percent would be required by the engine between Mach numbers of 1.0 and 2.0. At subsonic speeds, choking at the minimum flow area of the inlet determines the maximum air flow handled, and a continuously increasing minimum inlet area would be required with decreasing flight Mach numbers. Such extreme variations in inlet areas appear impractical and compromises in matching the inlet to the turbojet engine are necessary.

The effect on performance of the compromises involved can be demonstrated by considering the characteristics of fixed-geometry high-pressure-recovery inlets. Selecting an inlet frontal area corresponding to a pressure recovery of 0.95 and $M = 1.0$ (fig. 2(b)), for example, will result in a pressure recovery of only 0.72 at the face of the compressor for $M = 2.0$ (irrespective of the much higher peak pressure recovery the inlet alone might provide). Actually, the pressure recovery of 0.72 at $M = 2.0$ corresponds to a normal-shock-type inlet rather than a high-pressure-recovery-type inlet. The cause of the low pressure recovery is that the inlet frontal area is less than the stream-tube area required for an inlet pressure recovery of 0.85 at Mach number 2.0 and the pressure recovery must decrease to satisfy the engine (equation (2)). Physically, this loss in pressure recovery occurs through a normal shock in the subsonic diffuser with supercritical inlet operation. The attendant thrust penalties for engine B associated with this loss in pressure recovery are indicated in figure 3, which presents the percentage change in thrust for each percentage change in pressure recovery for Mach numbers from 0 to 2.0. These data indicate that a 1-percent change in pressure recovery will result in a thrust change of from 1.25 to 1.4 percent at supersonic speeds, and as high as 1.75 percent at subsonic speeds. The low pressure recoveries associated with an undersize inlet would thus result in large losses in engine thrust and in general should be avoided.

Selecting an inlet frontal or capture area corresponding to the requirements for $M = 2.0$ and 0.85-percent pressure recovery (fig. 2(b)) will avoid the penalties of the undersize inlet discussed previously. Such a selection will, however, result in air-flow spillage and additive drag at the lower supersonic free-stream Mach numbers because the inlet capture area will be greater than the free-stream tube area required by the engine. This spillage can occur behind an oblique shock, a normal shock, or an oblique-normal-shock combination, depending on the inlet design. The magnitude of the inlet drag penalties associated with air spillage are shown in figure 4 as a percentage loss in ideal engine thrust for a range of Mach numbers. The additive drag for oblique-shock spillage was calculated for the optimum compression angle (peak inlet pressure recovery)

at each Mach number. For a given percentage spillage at a particular
free-stream Mach number, the drag penalty behind an oblique shock is
only 17 percent of the drag penalty associated with spillage behind a
normal shock; for a given drag penalty, the amount of air which can be
spilled increases with decreasing supersonic Mach number. From drag con-
siderations, inlet design compromises should therefore be made at the
lower Mach numbers and the required spillage should occur behind an
oblique rather than a normal shock.

The preceding general discussion has described the inlet-engine
matching problem and has indicated that the design compromises will
affect the thrust and the drag of the configuration. The desirable com-
promise will be the condition for which the thrust minus the drag of the
inlet-engine combination is maximized over the operating range. A non-
dimensional thrust-minus-drag parameter, which is defined as the ratio
of engine thrust minus inlet drag divided by ideal thrust, has been
selected to evaluate the design compromises for a two-dimensional ramp-
type inlet. The drag used herein includes only the calculated additive
drag of inlet due to spillage of air. The engine thrust was calculated
by assuming a representative variation of engine pressure ratio (engine B)
with Mach number, an afterburner temperature of 3900° R, a re-expanding
exhaust nozzle, and operation in the tropopause. For subcritical inlet
performance a pressure recovery of 95 percent of the theoretical inlet
total pressure was used and for supercritical operation the pressure
recoveries were calculated from equation (2). The ideal engine thrust,
which is based on the same engine operating conditions presented pre-
viously, was calculated for an inlet pressure recovery of 100 percent.

The variation of the thrust-minus-drag parameter with free-stream
Mach number for various single-oblique-shock-type two-dimensional inlet
designs is shown in figure 5. The reference curve indicates the perform-
ance of an inlet which has zero additive drag and the assumed subcritical
pressure recovery at each Mach number. It therefore represents an inlet
design which has variable area and variable-geometry characteristics over
the Mach number range. As such, the reference curve represents the limit-
ing or maximum performance which would be obtained with the assumptions
used.

A fixed-geometry inlet designed for M = 2.0 will result in a thrust-
minus-drag loss of approximately 20 percent over most of the Mach number
range (fig. 5(a)). Although the frontal area of the inlet exceeds the
stream-tube area required at the lower supersonic Mach numbers, the oblique
shock generated by the compression surface moves ahead of the inlet lip
and results in excessive spillage and an entering stream-tube area less
than that required by the engine. In order to satisfy the engine air-
flow requirements, the engine therefore literally sucks the normal shock
down into the subsonic diffuser and causes large losses in thrust as a
result of the low pressure recoveries.

At subsonic speeds, the minimum area, which is considerably smaller than the frontal area of the inlet, limits the air flow. The large air flows required by the engine therefore cause choking at the inlet with internal acceleration and large losses in pressure recovery.

Selecting an intermediate design point such as $M_0 = 1.5$ reduces the pressure losses due to supercritical operation at the lower speeds. At Mach numbers above the design value, the capture area of the inlet with the oblique shock intersecting the lip is smaller than the stream-tube area required by the engine and supercritical inlet operation results. This inlet design reduces the thrust loss at Mach numbers below 1.7 but increases the losses at higher speeds. For example, at $M = 2.0$, a 15-percent penalty in thrust minus drag is incurred.

Selecting a minimum inlet area to provide the required air flow at a free-stream Mach number of 0.85 and an inlet velocity ratio of 1 while maintaining the inlet geometry (16° ramp) to obtain high pressure recovery at a Mach number of 2.0 results in an inlet which is capable of delivering air flow in excess of the engine requirements at Mach numbers above the design value. The air-flow spillage behind the oblique-normal-shock configuration results in thrust-minus-drag losses of 13 percent at a Mach number of 1.3 and 6 percent at a Mach number of 2.0.

None of the fixed-geometry designs which have been considered approaches the maximum thrust minus drag attainable over the Mach number range, except in a narrow range near each design Mach number selected. Consideration of the problems associated with the three fixed-geometry designs of this analysis indicates that an inlet designed for a free-stream Mach number of 2.0 would be most amenable to modifications. Reduction of the excessive-spillage characteristic of this inlet at the supersonic Mach numbers below the design Mach number could be accomplished by decreasing the ramp angle. This would have the additional advantage that a more nearly optimum ramp angle on the basis of pressure recovery could be attained at each Mach number.

The performance of an inlet designed for maximum pressure recovery and zero spillage drag at a Mach number of 2.0 and utilizing an adjustable ramp which varied from 16° at the design Mach number to 0° at subsonic speeds is shown in figure 5(b). Near maximum performance was attained throughout the Mach number range with the variable-geometry inlet. A maximum loss of only 2 percent at $M = 1.3$ indicates that use of this technique should provide nearly optimum inlet-engine matching characteristics.

The principles employed in the preceding analysis can be extended to the spike-type inlet. Although variations in cone angle would be impractical, the Langley laboratory has suggested that by a translation of the spike the variable-geometry features could be attained over the Mach number

range considered. An analysis of such a variable-geometry inlet indica-
ted maximum thrust-minus-drag losses of 3 percent at a Mach number of 1.2
for a 25° half-angle cone.

Operation of the inlet-engine combination at various altitudes and
under conditions other than standard NACA atmosphere will influence the
matching problem. A detailed discussion of these problems is considered
to be beyond the scope of this investigation because the compromises
required are dependent on the flight program and the structural limits
of the airplane or missile. Analysis of a reasonable flight program
indicates, however, that such additional requirements can be satisfied
by the variable-geometry inlet.

MODEL DESCRIPTION

A model of the forward part of the fuselage of a proposed supersonic
airplane powered by two turbojet engines was used to investigate the prob-
lems associated with inlet-engine matching over a range of Mach numbers.
The model included two ramp-type side inlets located symmetrically about
the vertical center line of a representative fuselage. These inlets were
canted downward 2° with respect to the fuselage to compensate for the
cruise angle of attack of 3° at an altitude of 35,000 feet. An internal
duct extending to the station corresponding to the compressor inlet was
included to provide the required subsonic diffusion ahead of each engine.
A photograph of the model installed in the 8- by 6-foot supersonic wind
tunnel and rolled 56° for schlieren observation is presented in figure 6.

The semicircular side inlets shown in detail in figure 7 utilized
two-dimensional compression ramps. Ram-type scoops having a height h
of 0.8 inch equal to the boundary-layer thickness δ_{bl} were used to
remove the boundary-layer air ahead of the inlets. An internal duct
capable of handling the estimated tail-pipe cooling air flow was included
in the boundary-layer-removal system for each inlet. Blunt well-rounded
inlet leading edges for high performance at subsonic speeds (fig. 7(a))
as well as sharp inlet leading edges for maximum performance at super-
sonic speeds (fig. 7(b)) were investigated. The length of forebody ahead
of the boundary-layer scoop was approximately 6.3 feet with a correspond-
ing Reynolds number of 29×10^6 at the supersonic Mach numbers.

Model construction details prohibited the use of the variable-
geometry-type inlet previously discussed. However, the variable-geometry-
type inlet including probable fairing details in the subsonic diffuser and
the straight inlet sides required by a movable ramp was simulated by con-
structing 14° and 6° compression fixed-angle ramp inlets. The 14° ramp
was selected for optimum performance at the local Mach number of approxi-
mately 1.83 ahead of the inlet, which occurred at a free-stream Mach num-
ber of 2.0, for this model. In a similar manner the 6° ramp was determined

to provide optimum performance at a free-stream Mach number of 1.5. A sketch showing typical cross sections and model dimensions of the several ramp and inlet geometries is presented in figure 8.

Total-pressure measurements were made at the station corresponding to the compressor inlet by means of 33 pitot tubes located in each duct. The average total pressure at this station was obtained from an area weighting and was used to calculate the mass flow based on the area of the choked exit.

Mass flows through the inlets and the boundary-layer ducts were varied by means of remotely actuated control plugs attached to the model sting. The three-component strain-gage balance located inside the model did not include the force on the control plugs and therefore only the internal-duct force, fuselage drag, and model-base force were recorded. The model base was pressure instrumented and the plug assemblies were surrounded by a metal shield to provide nearly uniform base pressures. The force on the shield was not recorded by the balance.

The investigation at a Mach number of approximately 0.63 was conducted by operating the 8- by 6-foot supersonic tunnel subsonically. For take-off (zero forward speeds), inlet-air-flow conditions were simulated by attaching the model discharge ducts to the tunnel exhauster equipment in such a manner that the air flow could be controlled by the exit plugs.

RESULTS AND DISCUSSION

The pressure recovery, drag, and engine thrust-minus-drag characteristics of the blunt-lip 14^{0} ramp inlet configuration are shown in figure 9 for Mach numbers from 1.5 to 2.0. The total-pressure recovery is presented as the ratio of the total pressure at the duct discharge P_2 to the free-stream total pressure P_0 and as such includes the supersonic and subsonic diffusion losses. The model fore drag presented is defined as the internal thrust minus the sum of the balance reading and the base drag. The internal thrust is defined as the change in momentum, from free stream to the diffuser-discharge station, of the air passing through the inlet. The engine thrust was calculated by using the assumptions mentioned in the section INLET-ENGINE MATCHING except that the experimentally determined values of presssure recovery were used.

The total-pressure recoveries for the 14^{0} ramp inlet increased slightly with decreasing mass flow and stable inlet flow was observed for all conditions of operation. Maximum pressure recoveries of 0.87 at a Mach number of 2.0 ($M_{canopy} \approx 1.83$) and 0.97 at a Mach number of 1.5 ($M_{canopy} \approx 1.39$) are in good agreement with values obtained for nose inlets (reference 2).

The model fore drags exhibit the characteristic rise with subcritical inlet operation because of the increase in additive drag. The minimum drag increased with decreasing Mach number because of the increased air spillage around the inlets as the oblique shock generated by the compression ramp moved ahead of the inlet lip. This is shown qualitatively by the schlieren photographs in figure 10.

It is not apparent from the data presented in figure 9(a) whether the inlet-engine matching points occur at the maximum thrust minus drag because of the simultaneous increase in pressure recovery and drag in the subcritical range of inlet operation. Therefore the engine thrust minus model fore drag was calculated for a range of mass flows at M_0 of 2.0 and 1.5 by assuming that the inlet and diffuser-discharge areas were adjusted to provide the necessary engine air flow at the required diffuser-discharge Mach number.

The inlet scale factors used in the calculations are expressed in terms of $A_p/A_{p,d}$ in figure 9(b). An approximate correction for the drag was included to account for the change in inlet size, although the magnitude of the correction was less than 2 percent in terms of the thrust parameter.

The maximum thrust minus drag at a Mach number of 2.0 occurred with slightly subcritical inlet flow and indicated that the increase in pressure recovery is relatively more important than the increase in drag due to the air spillage. As shown by the solid symbols and the dashed line, inlet-engine matching was attained for a Mach number of 2.0 at an inlet pressure recovery of approximately 84 percent with near-peak thrust minus drag. At a Mach number of 1.5, however, matching occurred at the extremely low pressure recovery of 82 percent because of the excessive air spillage around the inlet. As a result of the low inlet pressure recovery for inlet-engine matching at a Mach number of 1.5 as compared with peak pressure recovery of 97 percent, a loss in thrust of approximately 20 percent was suffered (fig. 3), resulting in performance considerably below peak thrust minus drag.

The analysis indicated that inlet-engine matching could be made to occur near maximum pressure recovery at a Mach number of 1.5 by increasing the mass flow captured by the inlet. As shown in figure 10(b) the shock from the 14° ramp is detached and stands well ahead of the inlet lip. Decreasing the wedge angle, therefore, should result in an attached oblique shock which falls closer to the inlet lip and thus increase the mass flow entering the inlet and decrease the spillage drag. Calculations indicated that a 6° ramp would provide inlet-engine matching at maximum thrust minus drag for $M_0 = 1.5$.

The pressure-recovery, drag, and thrust-minus-drag characteristics of the 6° ramp configuration are presented in figure 11. At $M_0 = 1.5$ inlet-engine matching occurs at a pressure recovery of approximately 96 percent but far enough into the subcritical region to result in a thrust minus drag slightly below the maximum possible because of the additive drag penalty. The design could probably be further improved by slightly increasing the wedge angle. This would decrease the mass flow captured by the inlet, causing the normal shock to move closer to the inlet lip for inlet-engine matching and thus reduce the additive drag because the spillage would occur behind an oblique shock rather than an oblique-normal-shock combination.

Although the 6° ramp configuration was designed to operate at a Mach number of 1.5, the inlet was investigated at Mach numbers up to 2.0 to evaluate the off design performance. Inlet-engine matching at the higher Mach numbers occurred at approximately peak pressure recovery, but the large air-flow spillages associated with the subcritical inlet operation resulted in performance appreciably less than the maximum thrust minus drag. Schlieren photographs for the 6° ramp inlet are presented in figure 12 for Mach numbers of 1.5 and 2.0.

The increased performance associated with the use of a variable-angle ramp over the range of Mach numbers is confirmed by the data presented in figures 9 and 11; however, the minimum drag of the 14° ramp configuration is considerably higher than that of the 6° ramp configuration. This difference may be explained qualitatively by comparing the inlet flow conditions shown in the schlieren photographs of figure 13. With maximum inlet air flow (minimum drag) the normal shock is located much farther ahead of the inlet leading edge for the ramp configuration of 14° than for that of 6° and results in additive drag due to air spillage. The increased air spillage for the 14° ramp configuration may be attributed to the higher internal contraction associated with the use of the blunt lip and to the higher flow angles at the inlet lip.

In order to eliminate the additive drag associated with the blunt subsonic inlet leading-edge design at supersonic speeds, a 14° ramp inlet with sharp leading edges for high performance at supersonic speeds was investigated. A cross-sectional view of the inlet is presented in figure 8(c). For inlet-engine matching of the sharp-lip design at a Mach number of 2.0, a decrease in inlet capture area was required to compensate for the reduced air spillage at critical operation. This decrease was accomplished by moving the position of the inlet leading edge while maintaining the position of the ramp.

The performance characteristics of the sharp-lip 14° ramp inlet are presented in figure 14 for a range of Mach numbers. Comparison of the inlet pressure recoveries with the data presented in figure 9 for the blunt subsonic inlet design, indicates good agreement. The minimum drag

for the sharp-lip inlet design, however, was decreased 27 percent as compared with the minimum drag for the blunt subsonic inlet configuration. Approximately 3 percent of this reduction in drag can be attributed to the decreased spillage behind the oblique shock due to the movement of the inlet lip. The other 24 percent reduction in drag, which appears to be associated with the decrease in inlet leading-edge bluntness, represents 7 percent of the ideal engine thrust at a Mach number of 2.0.

As indicated by the inlet-engine matching condition at a Mach number of 2.0, the sharp-lip design is slightly undersize and results in a thrust minus drag below the maximum attainable. The schlieren photographs presented in figure 15 reveal some air spillage behind a detached bow wave at the lip of the inlet which could not be accounted for in the calculations. A study of the inlet design indicated that the detached wave resulted from excessive turning of the flow along the internal surface of the cowling. The internal angle of 10° with respect to the canopy reference surface was required to match the existing ducts and to provide cowling strength. Increasing the frontal area of this inlet or redesign of an inlet not limited by the model geometry should result in maximum thrust minus drag.

The relative performance of the various inlet configurations and the performance penalties associated with operating fixed-geometry-type inlets over a range of Mach numbers are summarized in figure 16 in terms of the nondimensional thrust parameter.

As predicted by the analysis, operation of fixed-geometry inlets at off-design Mach numbers resulted in large performance penalties. For example, the 14° ramp blunt-lip configuration designed for M_0 of 2.0 ($M_d = 2.0$) resulted in a decrease in the thrust parameter at M_0 of 1.5 equivalent to approximately 20 percent of the ideal engine thrust when compared with the 6° ramp blunt-lip configuration designed for M_0 of 1.5 ($M_d = 1.5$). Conversely, operation of the $M_d = 1.5$ inlet at $M_0 = 2.0$ resulted in losses equivalent to approximately 7 percent of the ideal thrust compared with the $M_d = 2.0$ inlet. It should be pointed out that the magnitudes of the thrust parameters do not agree with the analysis, because the model fuselage drag was included in the calculations using the experimental data.

At a Mach number of 2.0, use of the sharp-lip rather than the blunt-lip 14° ramp inlet configuration resulted in an increase in the thrust parameter equal to 7 percent of the ideal thrust. However, the increase in performance associated with the lower drag of the sharp-lip inlet would be expected to decrease with decreasing Mach number.

The estimated performance of a variable-geometry-type inlet whose compression angle varies from 14° at M_0 of 2.0 to zero at M_0 of 0.63 is represented in figure 16 by the dash-dash-dot curve for a blunt-lip inlet

and the dash-dot curve for a sharp-lip inlet. The performance of the zero
ramp configuration at M_0 of 0.63 was obtained by extrapolating the
experimentally determined variation of pressure recovery with inlet mass-
flow ratio for the 14° ramp inlet operating at M_0 of 0.63 to account for
the reduced inlet mass-flow ratio associated with the increased inlet
area. This approximation indicates that engine-inlet matching with high
performance can be obtained for the Mach number range investigated with
an inlet whose geometry varies for supersonic and subsonic speeds.

Because one of the big advantages of a turbojet power plant is its
ability to provide thrust for take-off, the blunt- and sharp-lip 14°
ramp inlets were investigated at zero forward speed. The inlet charac-
teristics which are presented in figure 17 indicate that at the matching
diffuser-discharge Mach number pressure recoveries of only 67 and 74 per-
cent are available for the sharp-lip and blunt-lip designs, respectively.
These low pressure recoveries are due to inlet choking and can be alle-
viated by decreasing the ramp angle, which increases the minimum inlet
area. Accordingly, the experimental data were extrapolated to zero ramp
angle as discussed previously for the $M_0 = 0.63$ data (see fig. 16).
Inlet-engine matching now occurs at a pressure recovery of approximately
0.97 for the blunt-lip inlet and 0.86 for the sharp-lip inlet. The 11
percent loss in pressure recovery for the sharp-lip inlet represents
approximately 18 percent loss in thrust for the take-off condition and
may be prohibitive. Auxiliary inlets, adjustable translating cowl sec-
tions (reference 3), or rotating leading-edge cowl sections, however,
can be used to eliminate this penalty so that the sharp lip can be
available at supersonic speeds.

SUMMARY OF RESULTS

An analytical and experimental investigation of the problems assoc-
iated with the design of high performance inlets for a turbojet-powered
aircraft at Mach numbers from 0 to 2.0 was conducted. Two ramp-type
side inlets located symmetrically about the vertical center line of a
triangular shaped fuselage were investigated at a Reynolds number of
29×10^6 based on the length of forebody ahead of the inlets. For the
range of conditions investigated, the following general results are
indicated:

1. The wide range of air flows required by a turbojet engine oper-
ating from zero speed to $M_0 = 2.0$ resulted in operation off the peak
pressure recovery and minimum-drag operating points (critical points) of
a fixed-geometry-type inlet. Losses in thrust due to supercritical inlet
operation, additive drag penalties due to spillage of air around the
inlets, or some combination of these penalties over at least a part of
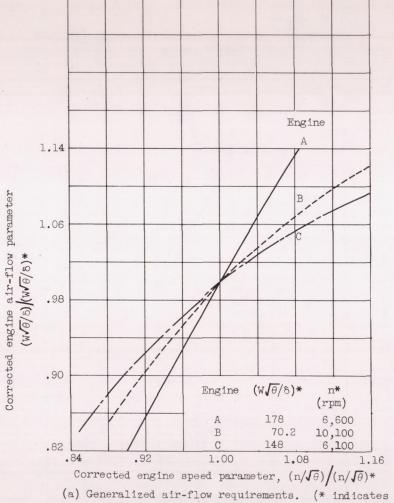the Mach number range were incurred.

2. Large thrust penalties incurred as a result of low inlet pressure recoveries (of the order of 1.25 to 1.75 percent for a 1-percent change in pressure recovery) indicated that high pressure recoveries should be maintained, even at the expense of some increase in inlet drag.

3. The analysis indicated that a variable-geometry-type inlet was required to provide inlet-engine matching with high performance. This was confirmed experimentally by investigating two inlets of different compression-ramp angles.

4. Experimentally it was established that with all the boundary layer removed ahead of the aft-inlets, total pressure recoveries of 0.97 at $M_0 = 1.5$ and 0.87 at $M_0 = 2.0$ which are comparable with those attainable with well designed nose inlets can be obtained.

5. Well rounded leading edges designed for high performance at subsonic speeds resulted in a drag increase equivalent to a 7 percent reduction in the calculated engine thrust at $M_0 = 2.0$ compared with a sharp-lip inlet. Use of the sharp-lip inlet at zero forward speed resulted in thrust losses of approximately 18 percent. However, this reduced performance could largely be eliminated while retaining the high performance at supersonic speeds by the use of auxiliary inlets or blow-in doors.

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    National Advisory Committee for Aeronautics
        Cleveland, Ohio
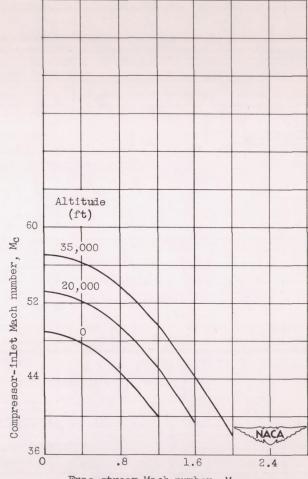
REFERENCES

1. Luskin, H., and Klein, H.: High Speed Aerodynamic Problems of Turbojet Installations. Douglas Rep. No. SM-13830, Douglas Aircraft Co. (Santa Monica), Sept. 1, 1950.

2. Brajnikoff, George B., and Rogers, Arthur W.: Characteristics of Four Nose Inlets as Measured at Mach Numbers Between 1.4 and 2.0. NACA RM A51C12, 1951.

3. Cortright, Edgar M., Jr.: Preliminary Investigation of a Translating Cowl Technique for Improving Take-Off Performance of a Sharp-Lip Supersonic Diffuser. NACA RM E51I24, 1951.

Engine

A

B

C

1.14

1.06

.98

.90

.82

Corrected engine air-flow parameter $(W\sqrt{\theta}/\delta)/(W\sqrt{\theta}/\delta)^*$

Engine	$(W\sqrt{\theta}/\delta)^*$	n^* (rpm)
A	178	6,600
B	70.2	10,100
C	148	6,100

.84   .92   1.00   1.08   1.16

Corrected engine speed parameter, $(n/\sqrt{\theta})/(n/\sqrt{\theta})^*$

(a) Generalized air-flow requirements. (* indicates
sea-level static NACA standard day values.)

Altitude
(ft)

60

35,000

52

20,000

0

44

36

Compressor-inlet Mach number, M_c

NACA

0   .8   1.6   2.4

Free-stream Mach number, M_0

(b) Compressor-inlet Mach number for
engine B; n and n*, 10,100 rpm;
compressor-inlet area of full-scale
engine, 1.94 square feet.

Figure 1. - Turbojet-engine characteristics.

(a) Diffuser-discharge Mach number; n and n*, 10,100 rpm; A_c/A_2, 0.71.

(b) Stream tube-area ratio; altitude, 35,000 feet; compressor-inlet area of model, 0.1215 square foot.
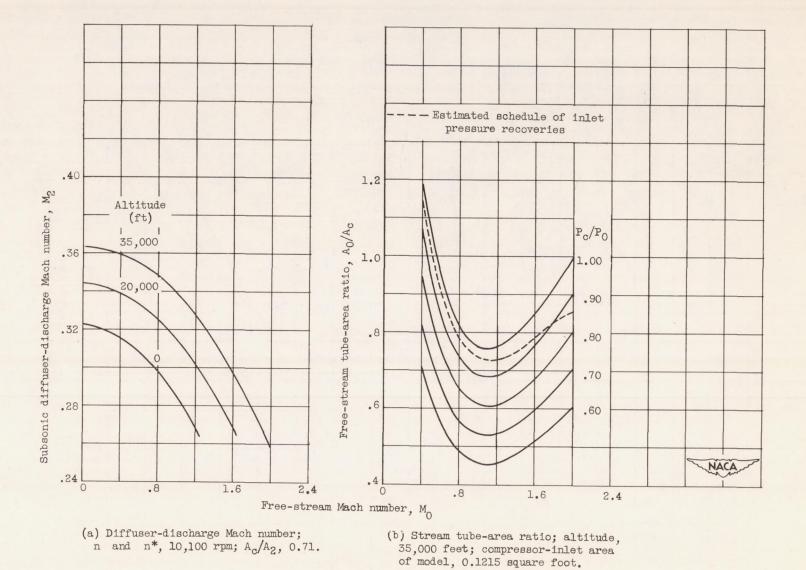
Figure 2. - Required inlet characteristics for inlet-engine matching for engine B.
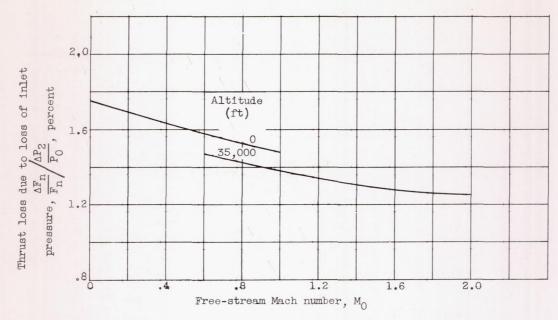
Figure 3. - Effect of inlet pressure recovery on engine thrust for two
altitudes and range of free-stream Mach numbers.
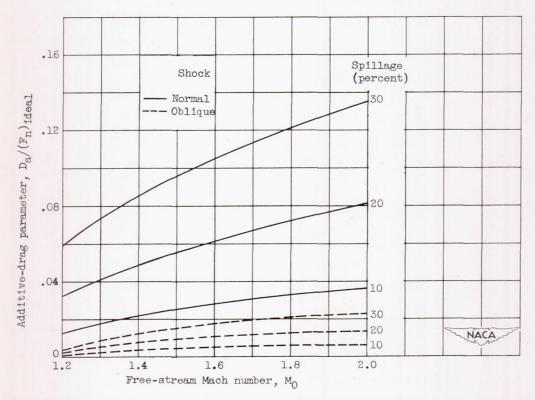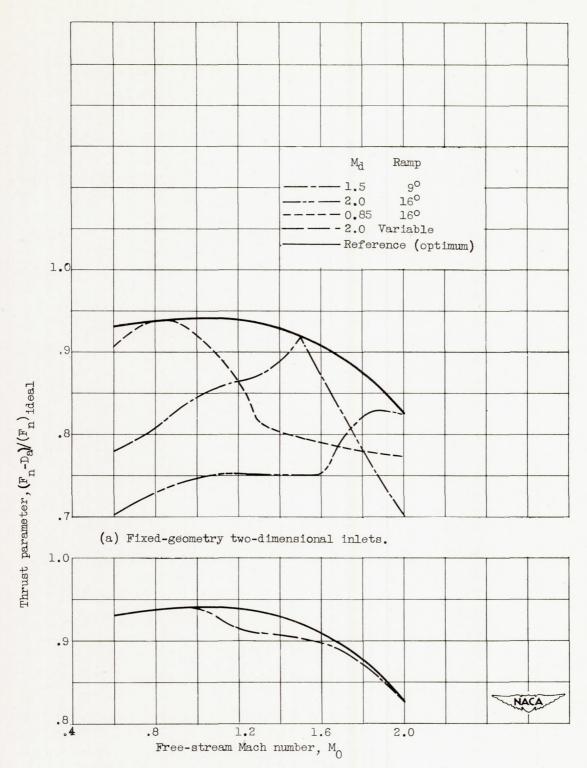
Figure 4. - Variation of additive-drag parameter with Mach number for
range of air-flow spillages behind oblique and normal shocks.

(a) Fixed-geometry two-dimensional inlets.

(b) Variable-geometry two-dimensional inlet.

Figure 5. - Thrust-minus-drag performance characteristics for several
inlet designs over range of Mach numbers from 0.6 to 2.0.
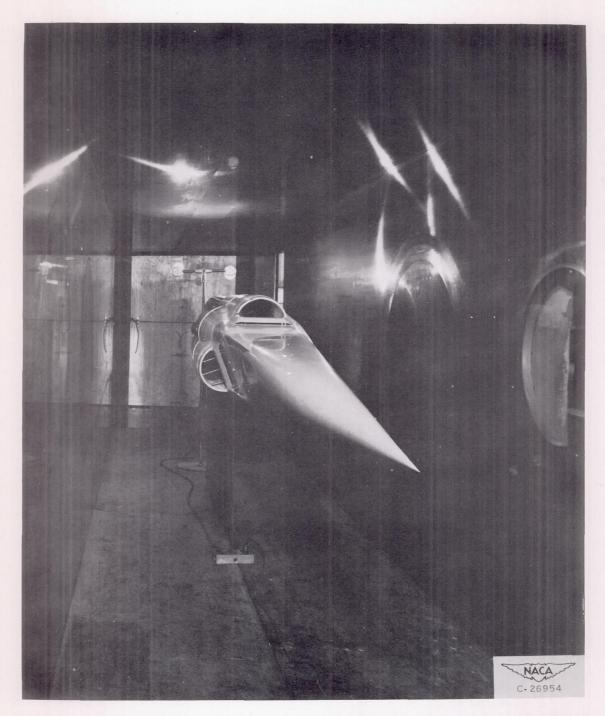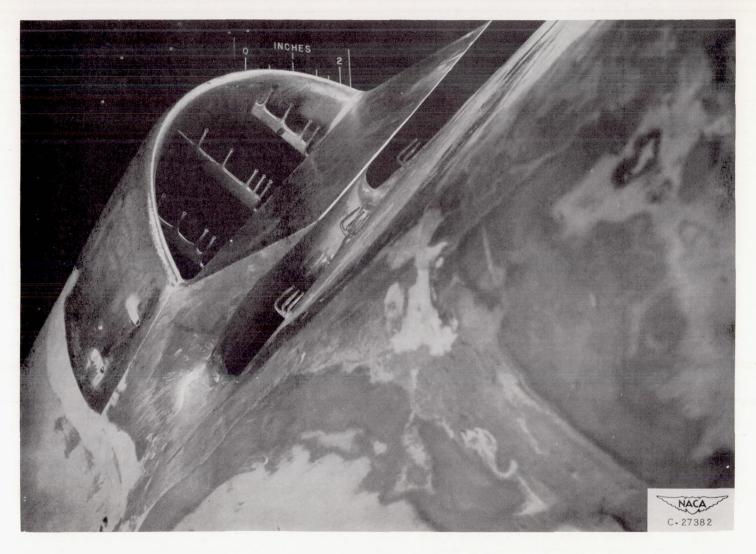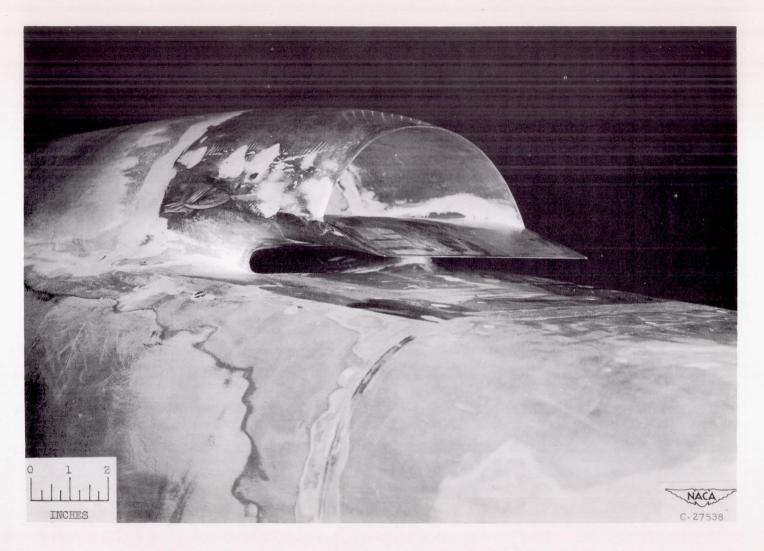
Figure 6. - Photograph of model installed in 8- × 6-foot supersonic wind tunnel. Model rolled 56° for schlieren observation.
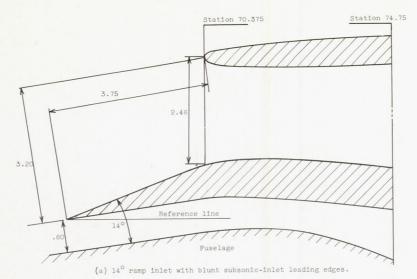
(a) Blunt subsonic-inlet leading edge.
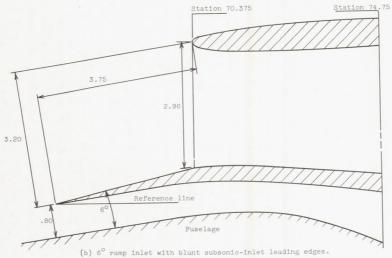
Figure 7. - Ramp-type side inlet mounted on modified triangular shaped fuselage; 14° ramp; height of boundary-layer scoop, 0.8 inch.

(b) Sharp supersonic-inlet leading edge.

Figure 7. - Concluded. Ramp-type side inlet mounted on modified triangular shaped fuselage; 14° ramp; height of boundary-layer scoop, 0.8 inch.

Station 70.375   Station 74.75

3.75

2.46

3.20

Reference line

14°

.80

Fuselage

(a) 14° ramp inlet with blunt subsonic-inlet leading edges.

Station 70.375   Station 74.75

3.75

2.96

3.20

Reference line

6°

.80

Fuselage

(b) 6° ramp inlet with blunt subsonic-inlet leading edges.

Station 69.125   Station 74.75

2.35

2.22

2.66

Reference line

14°

.80

Fuselage

(c) 14° ramp inlet with sharp supersonic-inlet leading edges.
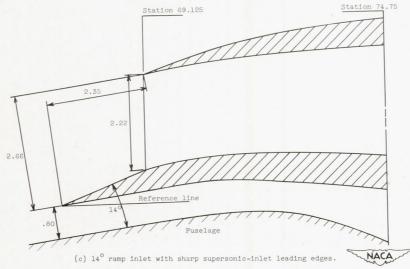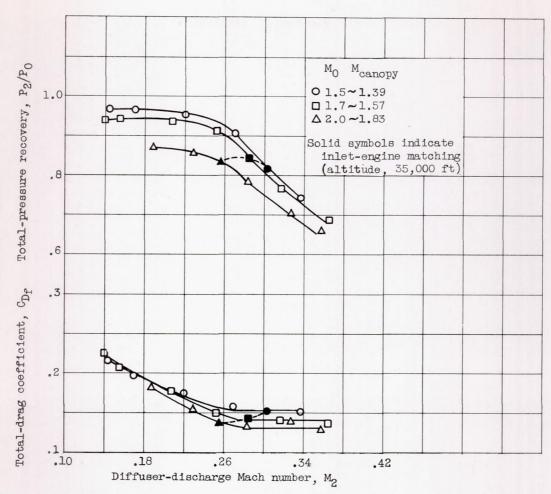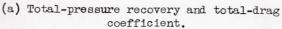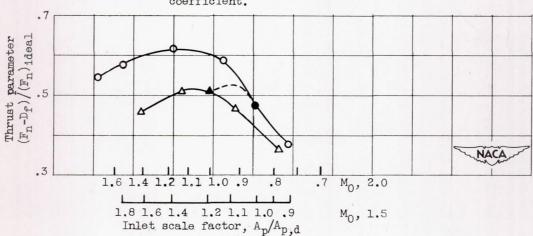
Figure 8. - Sketch of several two-dimensional ramp-type inlets showing typical cross
sections and model dimensions (all dimensions in inches).

(a) Total-pressure recovery and total-drag
coefficient.
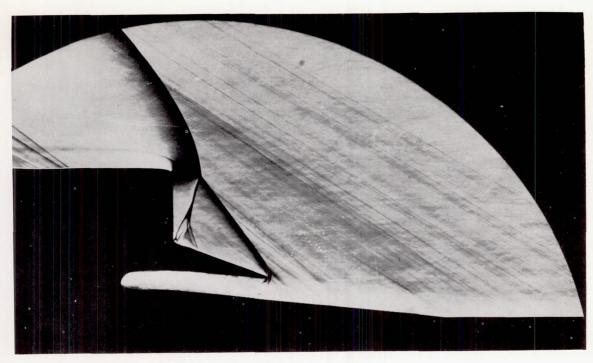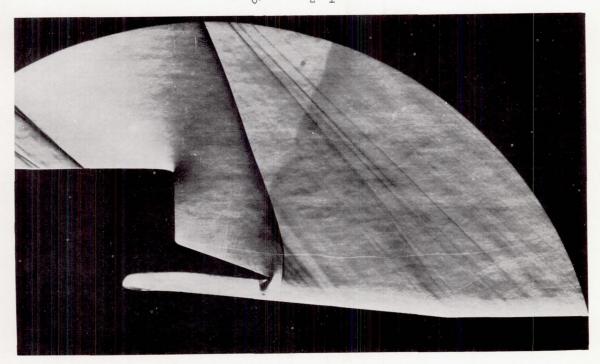
(b) Thrust parameter.

Figure 9. - Performance characteristics of subsonic-lip inlet with 14°
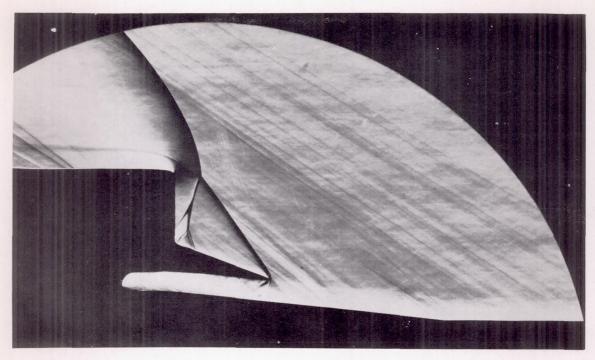ramp and boundary-layer removal (h/δ_{bl}, 1.0). Angle of attack, 3°.
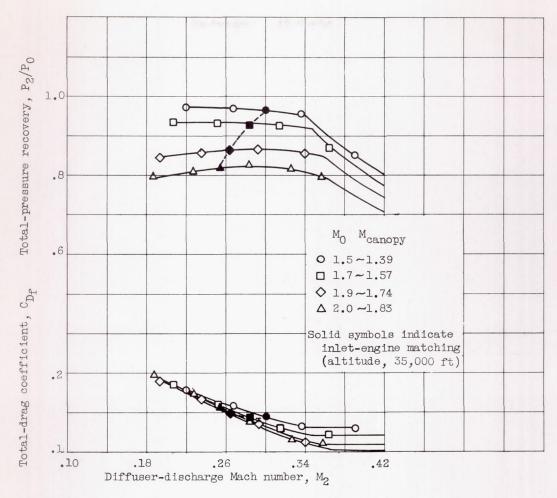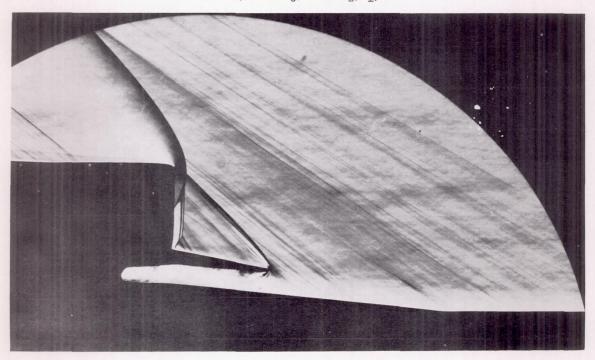
(a) M_0, 2.0; m_2/m_1, 0.613.

(b) M_0, 1.5; m_2/m_1, 0.524.

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Figure 10. - Schlieren photographs of subsonic-lip inlet for conditions of inlet-engine matching. $14°$ ramp with boundary-layer removal (h/δ_{bl}, 1.0).

(a) Total-pressure recovery and total-drag
coefficient.

(b) Thrust parameter.
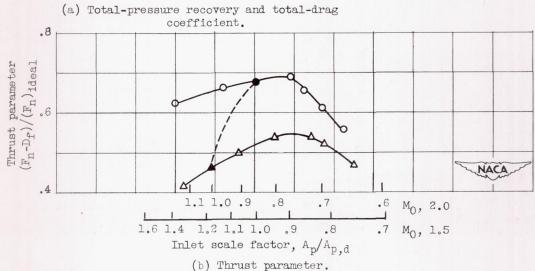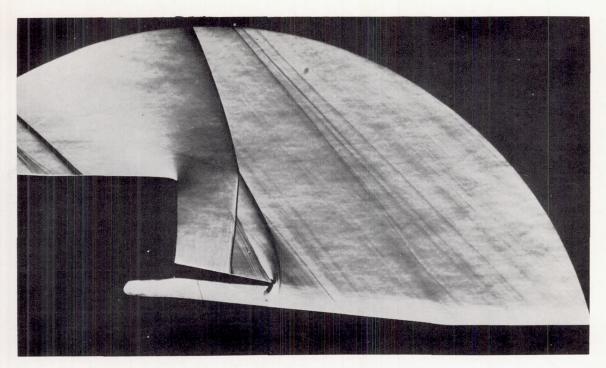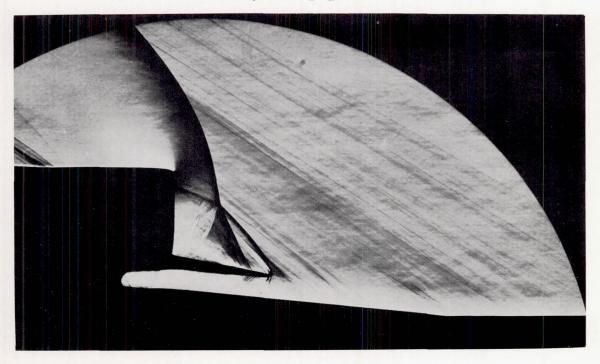
Figure 11. – Performance characteristics of subsonic-lip inlet with
6° ramp and boundary-layer removal (h/δ_{bl}, 1.0). Angle of attack, 3°.

(a) M_0, 1.5; m_2/m_1, 0.611.

(b) M_0, 2.0; m_2/m_1, 0.593.

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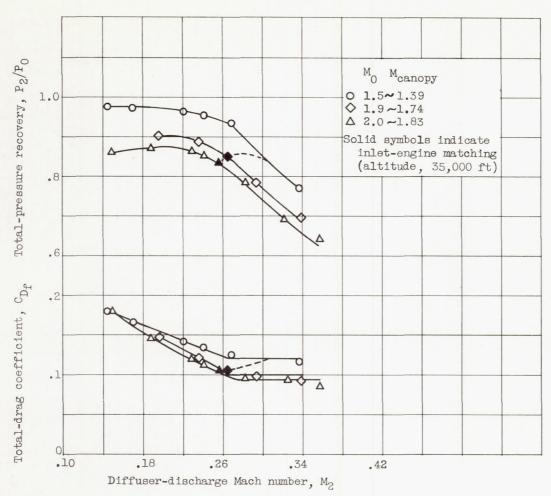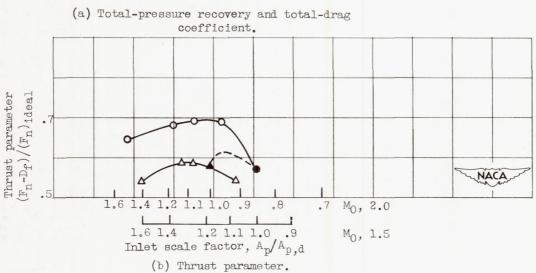Figure 12. - Schlieren photographs of subsonic-lip inlet with 6° ramp for inlet-engine matching. h/δ_{bl}, 1.0 .

(a) Ramp, 14°; M_0, 2.0; m_2/m_1, 0.652.

(b) Ramp, 6°; M_0, 2.0; m_2/m_1, 0.836.

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Figure 13. - Schlieren photographs of subsonic-lip inlet for conditions of maximum air flow. h/δ_{bl}, 1.0 .

(a) Total-pressure recovery and total-drag
coefficient.

(b) Thrust parameter.

Figure 14. - Performance characteristics of sharp supersonic-lip inlet
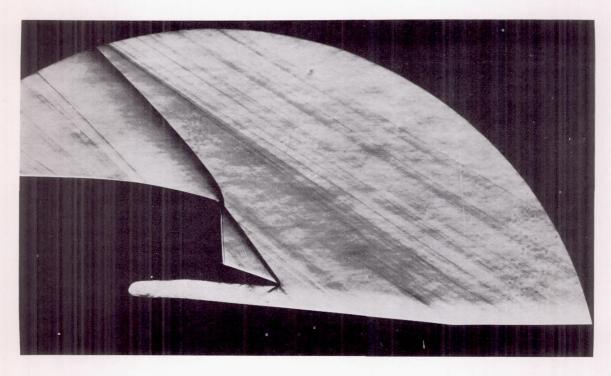with 14° ramp and boundary-layer removal (h/δ_{bl}, 1.0). Angle of
attack, 3°.

Figure 15. - Schlieren photograph of sharp supersonic-lip inlet for conditions of maximum air flow at Mach number 2.0 (m_2/m_1, 0.832). 14° ramp with boundary-layer removal (h/δ_{bl}, 1.0).
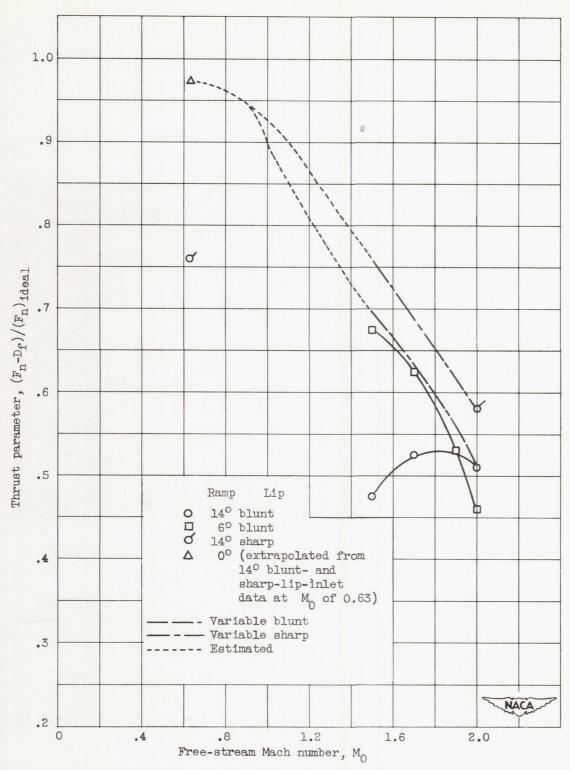
Figure 16. - Experimental thrust-minus-drag performance characteristics at inlet-engine matching for several inlet configurations over range of Mach numbers. Altitude, 35,000 feet; angle of attack, 3°.
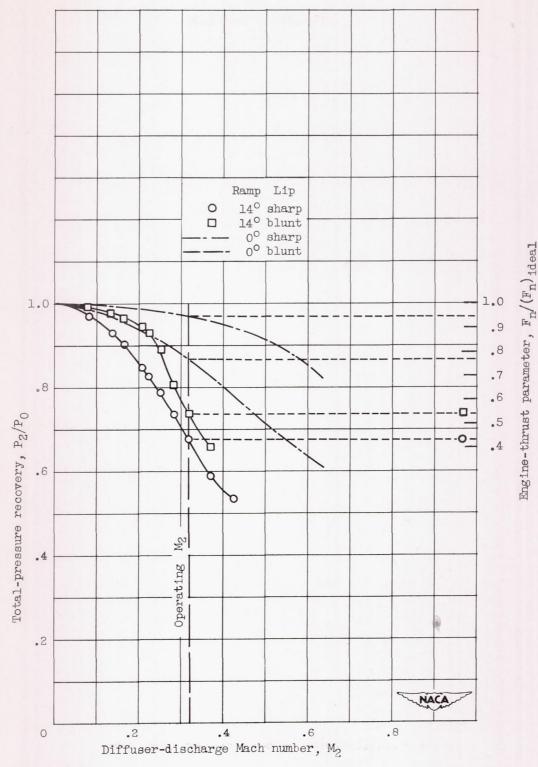
Figure 17. - Comparison of static take-off pressure recovery and thrust
    characteristics of inlet designs using subsonic and supersonic leading
    edges at sea level.

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